AD-A081 901 AIR FORCE INST OF TECH WRIGHT-PATTERSON AFB OH SCHOO--ETC F/G 21/5
TURBOJET THRUST AUGMENTATION BY SUDDEN EXPANSION TYPE AFTERBURN--ETC(U)
DEC 78 R DOTAN
UNCLASSIFIED AFIT/GAE/AA/78D-4 NL

1 OF 1
AD A
081901

END
DATE
FILMED
4-80
DTIC

AFIT/GAE/AA/78D-4

0

TURBOJET THRUST AUGMENTATION
BY SUDDEN EXPANSION TYPE
AFTERBURNER

THESIS

AFIT/GAE/AA/78D-4 Rami Dotan
Major IAF

AFIT/GAE/AA/78D-4

TURBOJET THRUST AUGMENTATION BY SUDDEN EXPANSION TYPE AFTERBURNER

Master's THESIS

Presented to the Faculty of the School of Engineering

of the Air Force Institute of Technology

Air Training Command

in Partial Fulfillment of the

Requirements for the Degree of

Master of Science

by

Rami Dotan

Major IAF

Graduate Aeronautical Engineering

Dece[redacted]78

Approved for public release; distribution unlimited.

Preface

As a part of a continuing study of thrust augmentation in small low cost turbojet, this study is intended to investigate the effects of length to diameter ratio of the afterburner, and air injection at the inlet to the afterburner on the overall propulsion system performance. This study also investigates the possibility of an eccentrically mounted afterburner operation and the influence of this configuration on the performance.

I wish to thank Dr. William Elrod for the direction and support he gave me as my thesis adviser. I would also wish to thank Mr. Leroy Canon and Mr. John Parks of the AFIT lab, for their assistance and support, and Mr. Carl Shortt, Mr. Dave Grube, Mr. John Brohas, Mr. Ron Ruley of the AFIT machine shop for their expert skills, craftsmanship and friendly support which resulted in the fabrication of the hardware required to conduct this study. I thank my wife, Michal, for the encouragement, patience, and understanding she showed me during the time of this study.

Rami Dotan

Contents

List of Figures

List of Tables

List of Symbols

Symbol		Quantity (Units)
Aug R	-	augmentation ratio
AR	-	aspect ratio (in.)
CPR	-	compressor pressure ratio
D	-	afterburner diameter (in.)
d	-	afterburner inlet diameter (in.)
F	-	corrected thrust (lb_f)
$\bar{F}$	-	uncorrected thrust (lb_f)
F_{sp}	-	specific thrust (lb_f-sec/lb_m)
FF_e	-	engine fuel flow (%)
FF_{ab}	-	afterburner fuel flow (%)
f_t	-	corrected fuel air ratio
$\bar{f}_t$	-	uncorrected fuel-air ratio
h	-	step height (in.)
Hr	-	hour
L	-	afterburner length (in.)
$\dot{m}_a$	-	corrected air mass flow (lb_m/sec)
$\bar{\dot{m}}_a$	-	uncorrected air mass flow (lb_m/sec)
$\bar{\dot{m}}_{ab}$	-	uncorrected afterburner fuel flow (lb_m/sec)
$\bar{\dot{m}}_{fe}$	-	uncorrected engine fuel flow (lb_m/sec)
$\bar{\dot{m}}_{ft}$	-	total uncorrected fuel flow (lb_m/sec)
$\dot{m}_{ft}$	-	total corrected fuel flow (lb_m/sec)

Symbol		Quantity (Units)
$\bar{N}$	-	uncorrected engine speed (rpm)
N	-	corrected engine speed (rpm)
P	-	pressure (psi)
P_t	-	total pressure (psi)
Δp	-	pressure differential (psi)
R	-	air gas constant (lb_f-ft/lb_m R)
SpGr	-	specific gravity
T	-	temperature (F)
T_t	-	total temperature (F)
ΔT	-	temperature differential (F)
TSFC	-	thrust specific fuel consumption (lb_m-Hr/lb_f)
θ	-	non-dimensional temperature correction factor
δ	-	non-dimensional pressure correction factor
η_T	-	augmenter thermal combustion efficiency (%)

Subscripts

a	-	air
A/B	-	afterburner
g	-	gage

Propulsion System Stations

0	-	ambient
3	-	compressor discharge
4	-	engine combustion chamber exit
5	-	engine turbine nozzle exit
6	-	A/B nozzle inlet

Abstract

An axisymmetric sudden expansion afterburner coupled to a small turbojet engine in various configurations, has been studied. The effects of length to diameter ratio variation on performance were investigated. The maximum thrust achieved was 62 pounds without afterburner augmentation, and 92 pounds with afterburner augmentation, each with turbine inlet temperature of 1700F.

Various methods of air injection into the combustion zone of the afterburner were examined, and found to be ineffective for purpose of thrust augmentation.

An eccentrically mounted afterburner with an eccentricity of 1.25 inch has been operated successfully. The performance of the propulsion system decreased in this configuration only by a negligible amount.

TURBOJET THRUST AUGMENTATION BY SUDDEN EXPANSION TYPE AFTERBURNER

I. Introduction

Background

A low cost propulsion system for utilization on a RPV is the main motivation for this investigation. An industrial turbocharger, AiResearch T18A-E, was converted to a small turbojet engine by adding a combustion chamber and turbine exit nozzle to the original compressor-turbine assembly (Ref. 8). The thrust levels reached by this configuration were too low for potential application, so thrust augmentation was considered.

Augmenting thrust by increasing turbine inlet temperature beyond that recommended by the manufacturer, leads to rapid destruction of the turbine blades (order of 3-5 minutes for 150F increase above the 1700F in turbine inlet temperature).

Thrust augmentation by means of afterburning appears to be more promising, but problems of size and weight of the entire propulsion system, make it necessary to minimize the dimensions of the afterburner while at the same time

increasing the combustion efficiency and thrust augmentation ratio. This must all be accomplished without using complicated parts that will increase propulsion system cost.

The Afterburner

An afterburner (A/B) has to provide a flame holding region and be of sufficient size for the combustion to be completed. A dump combuster, shown schematically in Fig. 1 is a simple device that provides the necessary conditions. Basically the dump combustor is a cylindrical tube with a sudden increase in diameter, which creates a step. A recirculation zone is established behind the step creating a flame holding zone. In addition, turbine exit swirl causes rotation of the whole recirculation flow, circumferencially around the A/B longitudinal axis. Even without the turbine exit swirl, the nature of flow after a sudden expansion includes rotational motion around the longitudinal axis (Ref. 4). The length of the A/B must be sufficient for full development of the flame and completion of the internal combustion. Any additional length beyond the minimum required, produces internal drag and increases dimensions and weight. The flame holding region behind the step can be improved by secondary air injection and the use of flame holders.

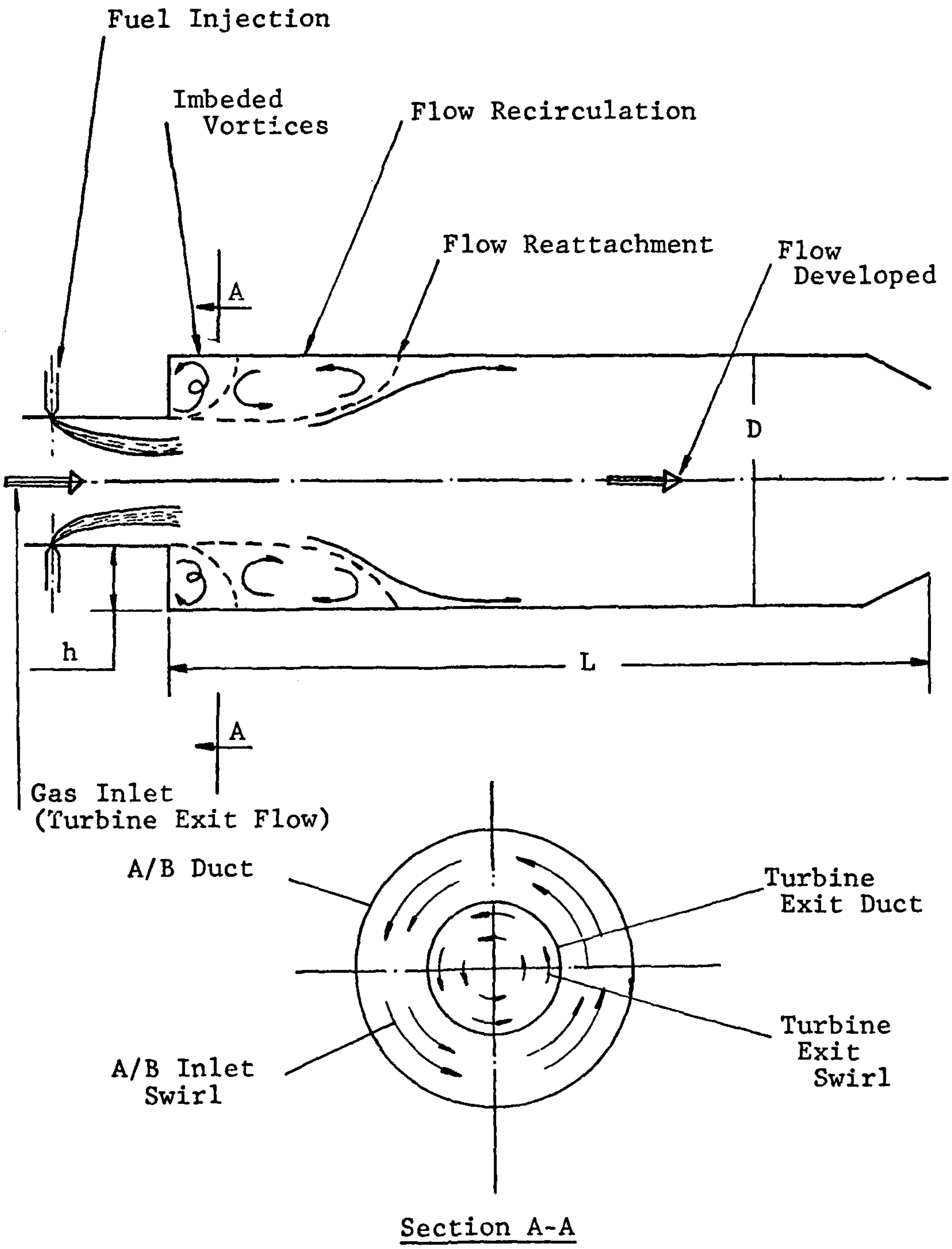


Fig. 1 Schematic of Dump Combustor Flowfield

Objectives

The objectives of this investigation were to:

1. Study the effect of A/B length to diameter ratio (L/D) on the propulsion system performance.

2. Investigate the possibility of enhanced mixing at the recirculation zone by air injection behind the step, and find the effects of this injection on the overall performance.

3. Study the effect of eccentric entry of the engine gasses into the A/B. There is a possibility of an off-center installation of the A/B, due to structural considerations of the RPV. The questions are: is it possible to operate the A/B in this configuration; and what will be the effect of such an installation on the overall performance of the propulsion system.

Scope

The investigation was concerned with the engine-A/B configuration overall performance, while trying to evaluate the contribution of the A/B to the performance. The evaluation of A/B performance alone, is straightforward when it concerns fuel-air ratio, and temperatures or efficiency. It is not possible with the existing test apparatus, to determine the portion of the thrust contributed by the A/B, the A/B TSFC or specific thrust.

Turbine inlet temperatures were maintained at 1700F to insure acceptable operation life of the engine, and to have a safety margin in case of turbine inlet temperature oscillations due to the A/B lighting or fuel flow change and the instability they create. The A/B was water cooled to assure its survivability. Previous L/D performance investigations (Ref. 1) indicated that L/D=3 showed good performance. In this study, L/D=2.5 and L/D=3.5 were examined to determine if L/D=3 provides a local peak in performance.

II. Test Apparatus

Engine and A/B

The engine used was basically a modified AiResearch T18A-E turbocharger, to which a combustion chamber incorporating an MA-1A (air starter unit) combustion liner has been added. Turbine housing aspect ratios as defined in Appendix A, used were 1.7 (engine A) and 1.5 (engine B). Compressor housing aspect ratio used was 0.96. The A/B was made in modules, each having a suitable L/D so that various combinations of the whole A/B length to diameter ratio could be assembled. Each module was cooled by water flowing in a jacket that surrounded the inner diameter cylinder of the module (Fig. 2-3, Table I).

Table I
Afterburner Dimensions

Dimension	Inches
d	3.875
D	6.375
L	31.875 max.
	15.9375 min.
X	4.50
h	1.25
K	2;1.5

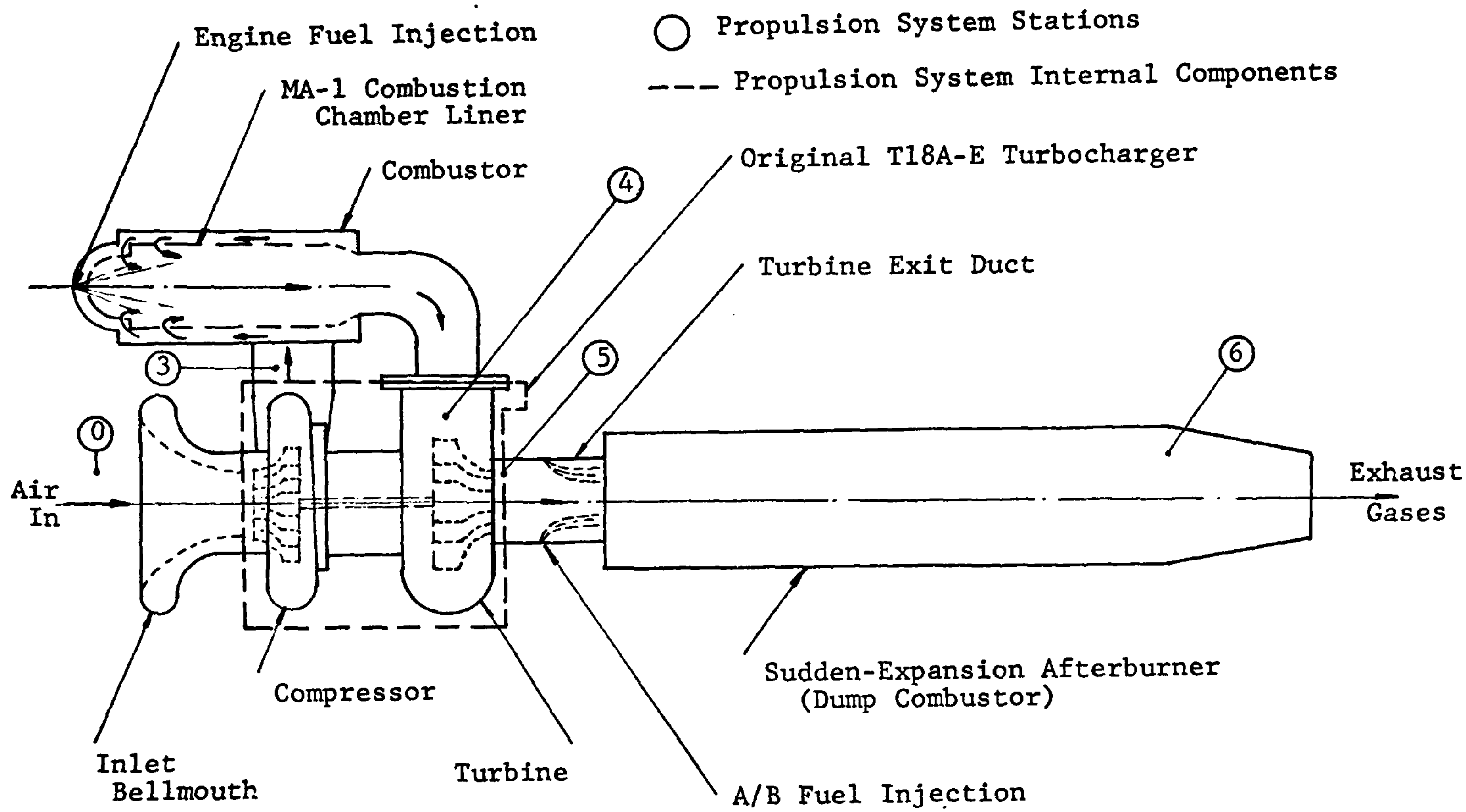


Fig. 2 Engine and Afterburner Schematic

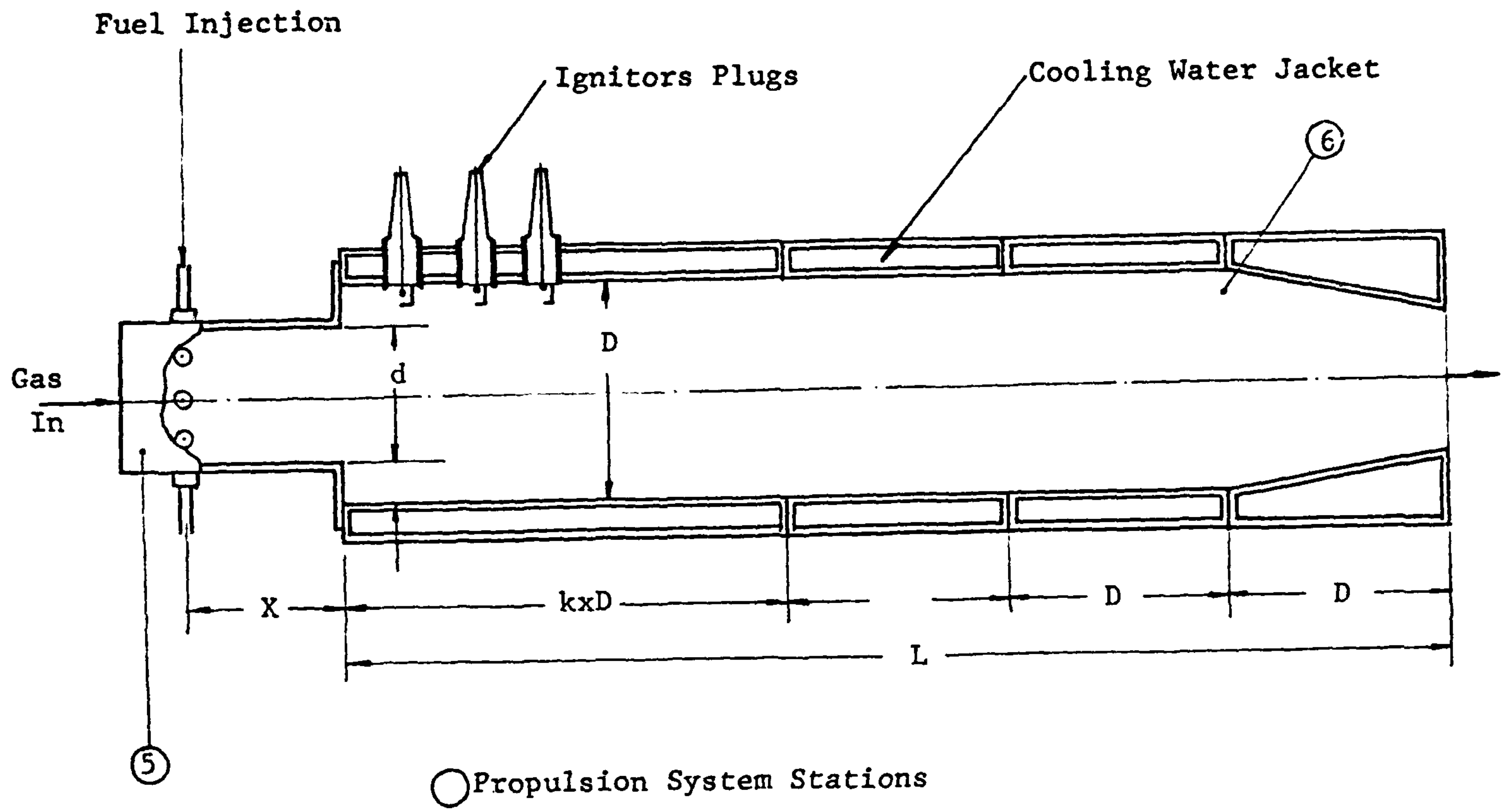


Fig. 3 Water Cooled Afterburner Assembly

The sudden expansion in the A/B, is created by the difference between the engine turbine exit duct inner diameter, and the A/B inner diameter. A/B exit nozzle was a two piece device, part of which was ejectable. This provided an area change to approximately equalize the A/B back pressure during non-operating and operating modes of the A/B. In the non-afterburning mode the nozzle diameter was 3.25 inches, and in the afterburning mode the nozzle diameter was 4 inches. Further description of the test cell systems and the engine can be found in Refs.1 and 2.

Air Injection System

High pressure bleed air from the compressor was used for injection at the step to increase mixing in the recirculation zone. This enabled examination of the air injection effects on the overall propulsion system performance.

At the step, air was injected through a manifold incorporating a single outer ring, and several inner injection rings. Each injection ring had six simple injectors, each one having a 0.120 inch diameter injection hole, which directed the air flow in three basic directions. The first direction was circumferential at the step cross section, clockwise (CW) and counter clockwise, (CCW), at two different step heights (Fig. 4a, b, e, g). Second, the injected air was directed axially in the direction of the

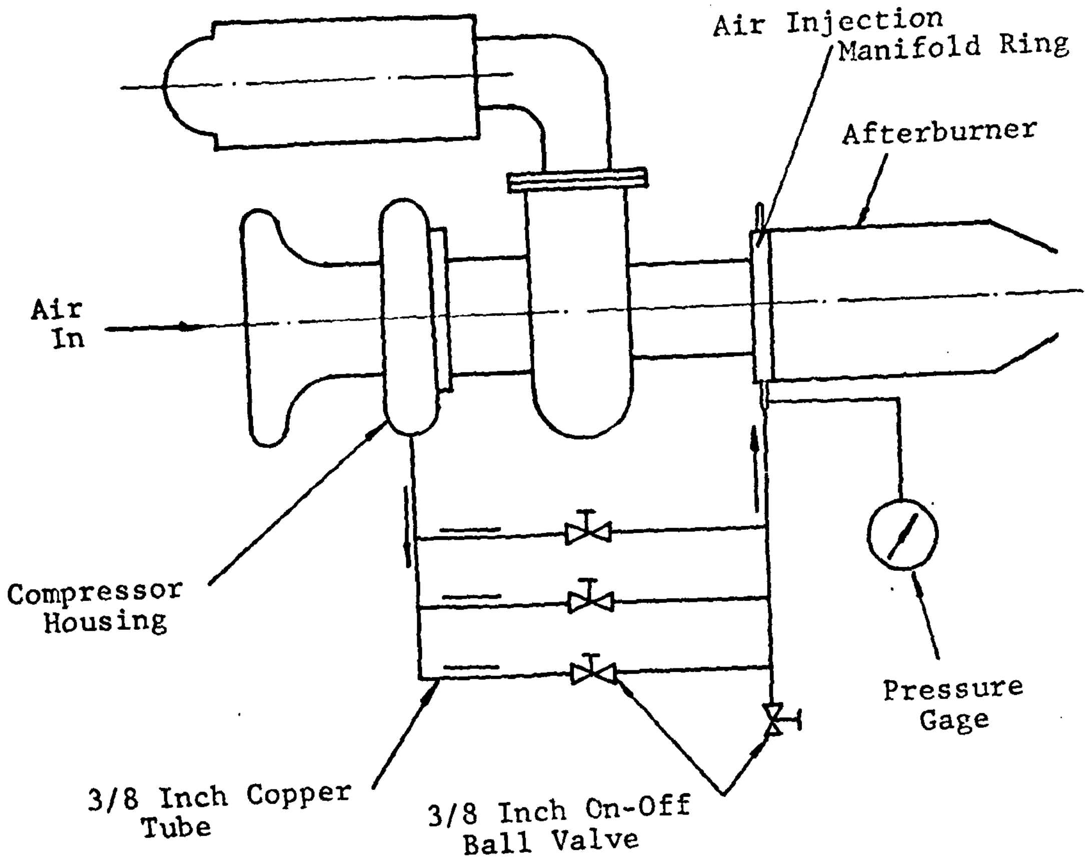
main flow, at one step height (Fig. 4a, c, f, h). The third direction was into the incoming flow to the A/B, at a 30 degrees angle (Fig. 4a, d, f, i).

By injecting air in the circumferential direction, the influence of the turbine exit swirl on the combustion was examined. The turbine exit swirl was CCW looking from the rear. Injecting air CW has an effect of retarding the swirl, and injecting air CCW has an effect of increasing the swirl. Injecting air in the axial direction influences the recirculation velocity, and injecting air into the incoming flow creates a bigger recirculation zone, as shown in Fig. 4d.

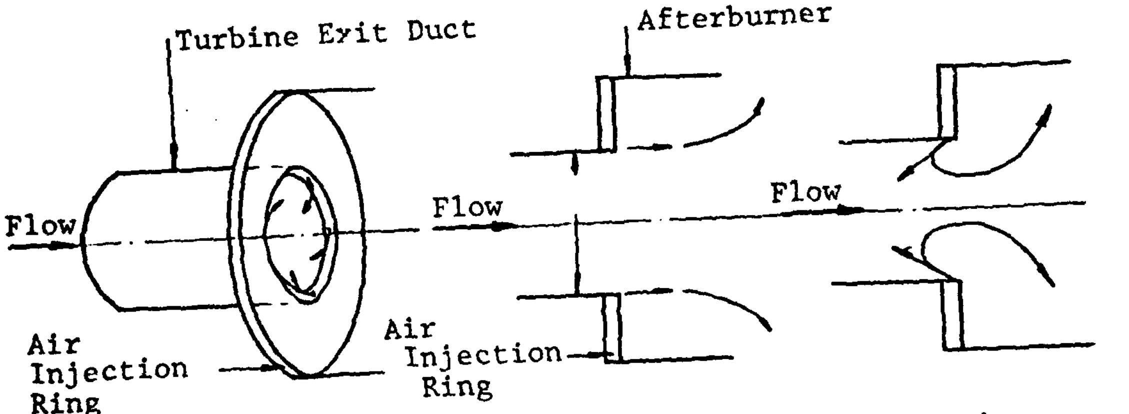
The injected air was transferred from the compressor housing by three 3/8 inch diameter copper tubes each having an on-off valve to permit variation of the flow. Air injection pressure was measured at the manifold inlet for the purpose of mass flow calculation as shown in Appendix A.

Eccentrically Mounted A/B

The A/B was coupled to the turbine exit nozzle by a single adapter plate, so that the upper part of the A/B inner diameter was flush with the turbine exit duct inner diameter as shown in Fig. 5.



a. Air Injection Manifold

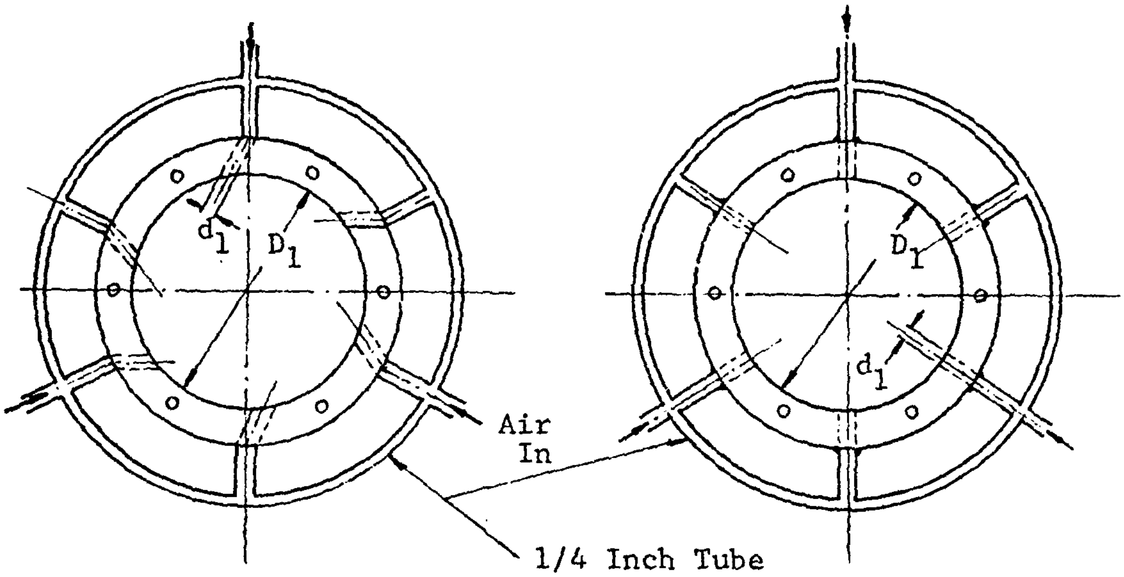


b. Circumferential Injection at two Different Step Heights. CW and CCW.

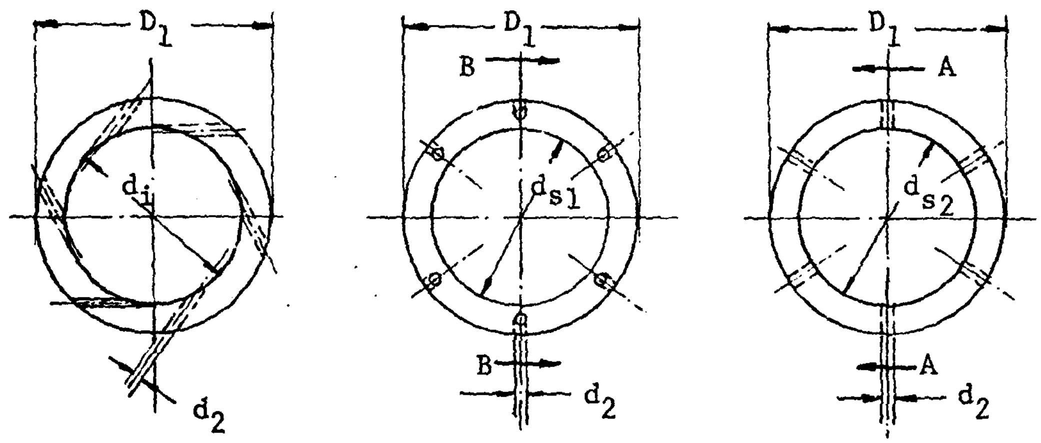
c. Axial Injection in the Direction of the Incoming Flow.

d. Injection into the Incoming Flow.

Fig. 4 Air Injection System Schematic



e. Circumferential Manifold Ring   f. Axial Manifold Ring

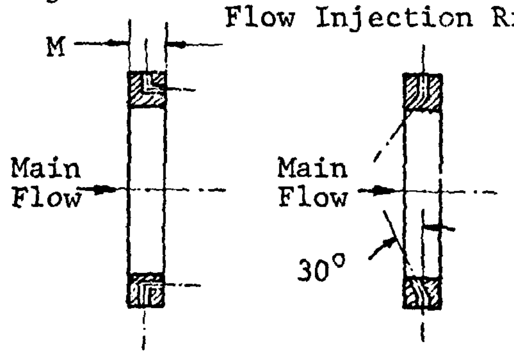


g. Circumferential Injection Ring CW or CCW

h. Axial Injection Ring

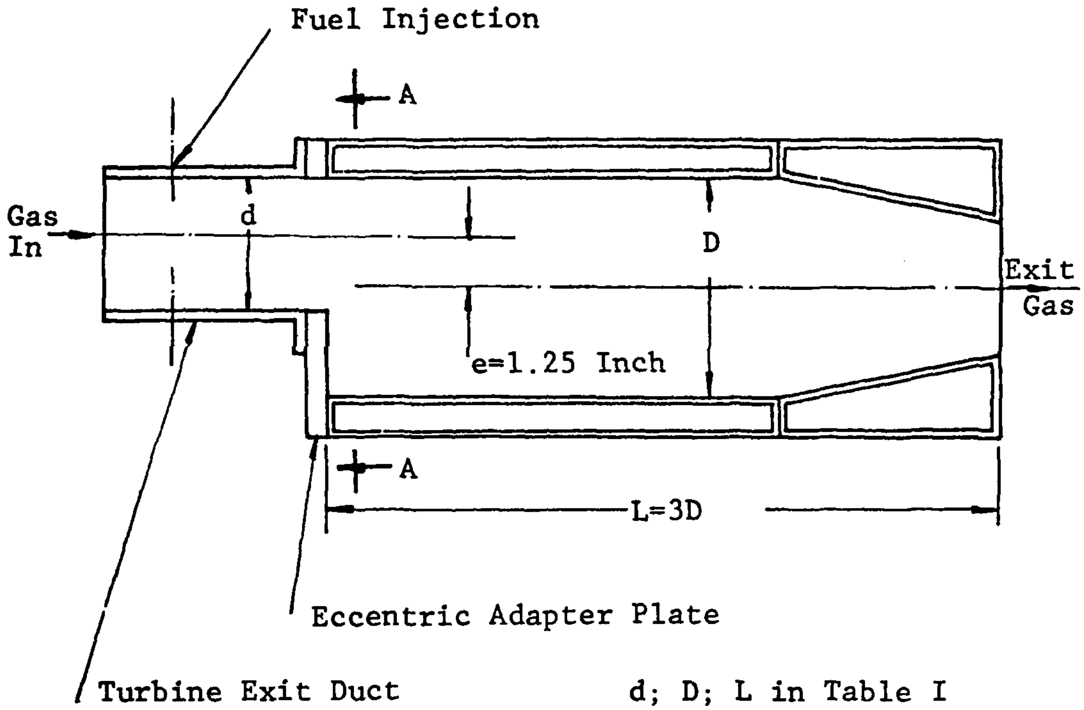
i. Into the Incoming Flow Injection Ring

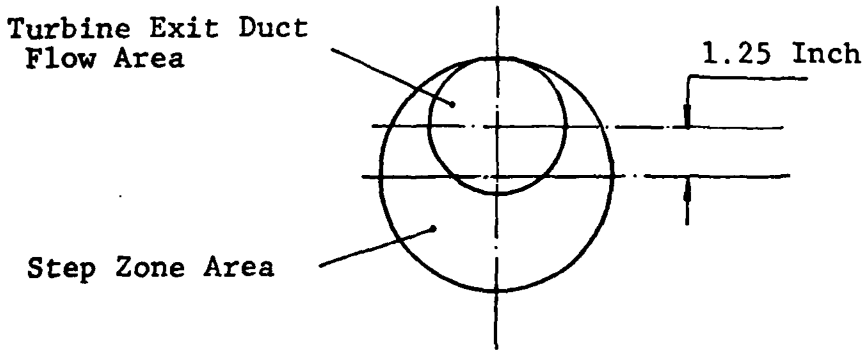
D_1	6.500	Inch
d_1	0.250	Inch
d_2	0.120	Inch
d_i	6.0;5.0	Inch
d_{s_1}	3.875	Inch
M	0.500	Inch
D_d	3.875	Inch
d_{s_2}	5.0	Inch



Section B-B   Section A-A

Fig. 4 Air Injection System Schematic (Cont')





Section A-A: Looking from A/B Rear

Fig. 5 Eccentrically Mounted A/B

Instrumentation

Engine pressures P_3; P_4; P_5; P_{t5} were measured by using pressure transducers, and recorded with a Honeywell 906B visicorder. A/B nozzle inlet pressure P_{t6} was measured by a direct reading pressure gage.

Engine temperatures T_{t3}; T_{t4}; T_{t5}; and A/B temperature T_{t6} were measured by bare wire thermocouples. T_{t3} and T_{t6} were recorded with the 906B visicorder; T_{t4} and T_{t5} were recorded manually from direct reading gages.

The overall thrust of the propulsion system was measured by a strain gage device and recorded with the 906B visicorder. Engine speed was measured by a light sensitive diode device which provided frequency reading proportional to rpm on a digital counter. This was recorded manually, and converted to rpm.

Fuel flow rates to the engine and A/B were measured by two turbine type flow meters which produce a frequency signal proportional to the flow rate. The signals were recorded with a Honeywell chart recorder, Model SY153X(28).

Air mass flow rate was determined from pressure measured by a water manometer connected to a measuring manifold mounted on the engine inlet belmouth. The air pressure was converted to air mass flow by a calibration chart supplied with the belmouth.

T_∞ was measured by a mercury type thermometer immersed into the engine inlet air flow, and recorded manually. Fuel

specific gravity was measured by a Hydrometer and recorded manually before every run.

Starting and ending time of each run were reported for the purpose of total running time records and engine life estimation.

All test cell instrumentation were calibrated before the experimental program started. Accuracy of each individual measurement is estimated according to the following:

1. Pressure (gage) ±2 psi
2. Pressure (visicorder) ±0.1 psi
3. Low range temperatures (0÷400F; visicorder) ±5F
4. High range temperatures (1150-3200F; visicorder) ±40F ; (gage) ±20F
5. rpm ±150
6. Thrust ±1 lb_f
7. Fuel flow rate $\pm 10^{-4}$ lb_m/sec
8. Air mass flow ±0.05 lb_m/sec
9. Ambient pressure ±0.01 in Hg
10. Ambient temperature ±1F
11. Fuel specific gravity ±0.005

III. Testing Procedure

Investigation of ramjet dump combustor has showed that the optimal length to diameter ratio of the combustor, needed for good mixing and best performance is $L/D \cong 5$ (Ref.10). Work done by Barham (Ref. 1) showed that a combustor having L/D=3 might perform as well as one having L/D=5. This investigation considers performance of combustors having L/D from 2.5 to 5, with particular interest in the range of 2.5 to 3.5. In these studies the step height was 1.25 inches. In addition, air injection of the step zone in order to create better mixing was studied, at 1.25 inch and 0.75 inch step heights. Another study was made concerning an eccentrically mounted afterburner, with eccentricity of 1.25 inches (Fig. 5). The actual running procedure used was identical for all test configurations. Many configurations were tested more than once to insure repeatability of data. Basically the engine was started, warmed up, and the throttle advanced until the turbine inlet temperature (T_{t4}) reached 1700F (defined as MIL 1). The engine was allowed to stabilize at this setting before recording MIL 1 data points. Following this, the A/B was ignited, the nozzle exit area was changed by ejecting part of the nozzle and the engine retrimmed to 1700F. This condition provided the second data point. Additional data

points were obtained by increasing A/B fuel flow in increments of 10% until reaching 100%. At each data point, turbine inlet temperature was retrimmed to maintain 1700F. At the conclusion of A/B run, the A/B fuel flow was cut off and the turbine inlet temperature reset to 1700F. The final data point (MIL 2) was recorded with the ejectable nozzle removed.

IV. Experimental Results

The experiments were performed using engine A and engine B. Augmentation ratio is defined as augmented thrust divided by MIL 1 thrust. Data reduction methods and calculations are shown in Appendix A. Data points in the graphical presentation of results are connected by straight lines for the purpose of clarity and not to indicate measured parameter performance between the points.

Effects of L/D Variation

The effects of L/D variation on A/B performance, are presented in Fig. 6. In general, the experimental results confirm that a dump combustor type of augmenter having L/D=3 has the same performance as an augmenter having L/D=5, as shown in Ref. 1. Figure 6 gives a comparison of the best performance points for each L/D. The augmentation ratio of L/D=3 is higher than that of L/D=5, the efficiency is higher, T_{t6} is higher and TSFC is lower.

However, a major difference from previous work is found when analyzing the augmentation ratio versus TSFC or fuel-air ratio (f_t). It can be observed from Figures 7 and 8 that for a large increase in TSFC, there is a modest rise in thrust and augmentation ratio.

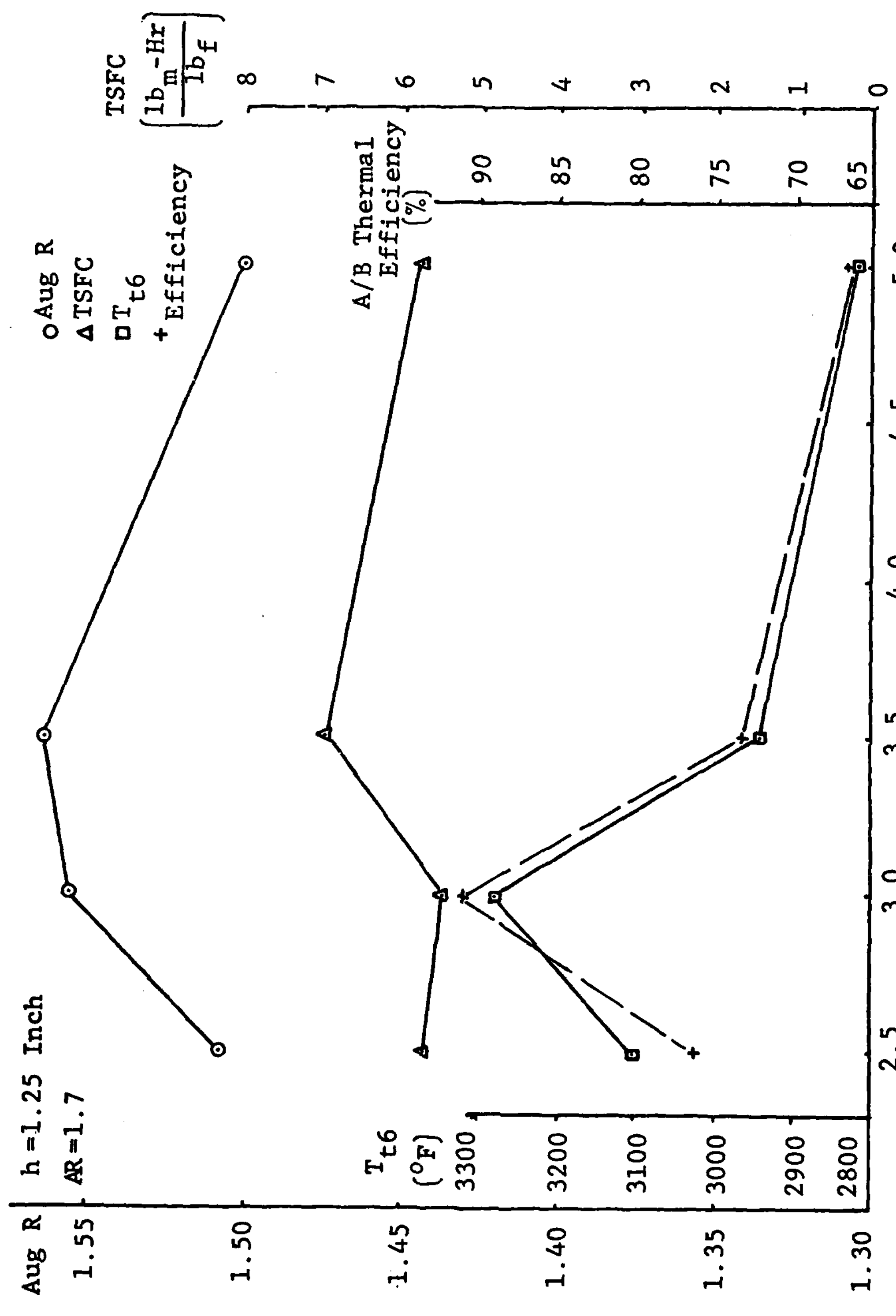


Fig. 6 Effects of L/D Variation on Aug R, TSFC, T_{t6} and Efficiency. Best Performance Data Points

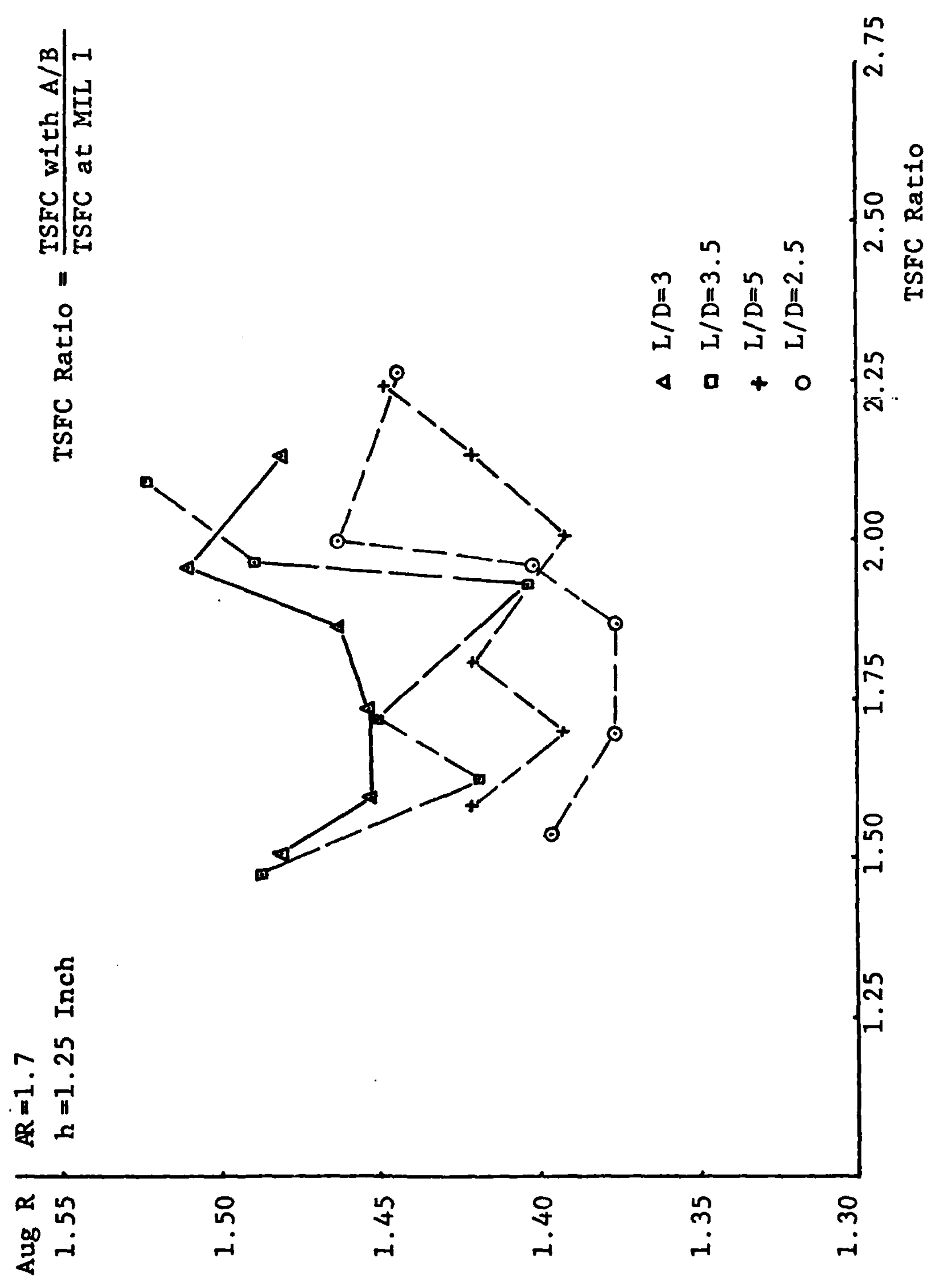


Fig. 7 Aug R vs TSFC Ratio at Various L/D

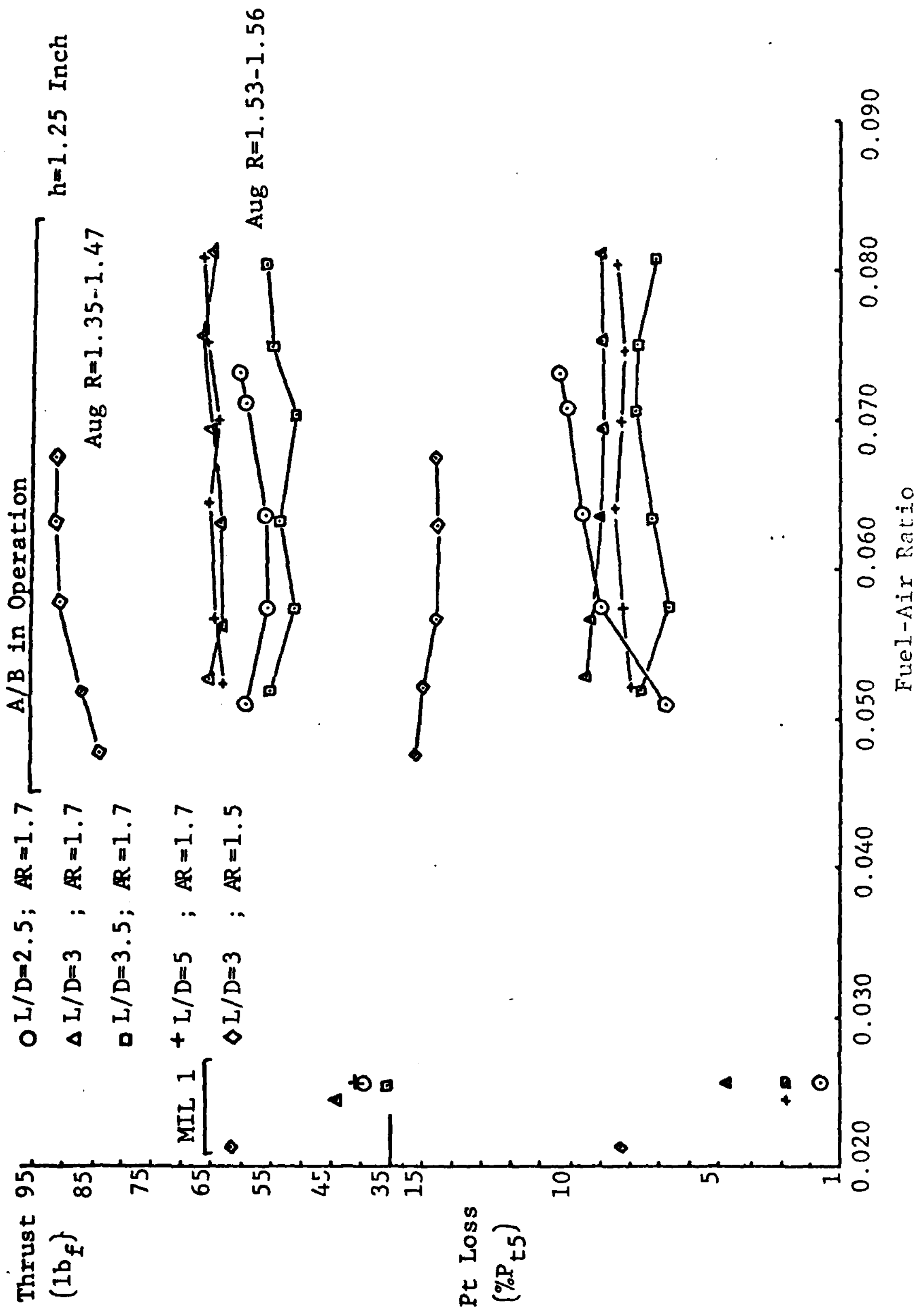


Fig. 8 Effects of L/D Variation on Thrust and Pressure Loss

Figure 7 shows that in the case of L/D=3 an increase of 30% in TSFC resulted in an increase of 2% in augmentation ratio. This phenomenon is also typical for the other L/D ratios presented in the graph. This led to an additional thrust and pressure loss pattern study, the results of which are presented in Fig. 8, for engines A and B.

It can be observed that although there is a change in absolute values of thrust and pressure loss between the two engines, the pattern of moderate change in thrust and pressure loss with increase of f_t is the same for all L/D ratios and for both engines. This suggests that an on-off operation mode of the augmenter at the lowest possible TSFC or f_t is desirable. Furthermore, the augmenter starting point performance with L/D=3 is better than that of L/D=5, although it's fuel consumption is 5% higher, as shown by Fig. 9.

Concerning the investigation around L/D=3 (i.e. L/D=2.5 and L/D=3.5), it was found that the performance of L/D=3 is better than that with L/D=3.5. Although the augmentation ratio achieved with L/D=3.5 was 0.5% higher than that of L/D=3, the TSFC of L/D=3.5 was 14% higher. Considering the increase in weight of nearly 17% (based on same diameter but higher length), the possibility of using the L/D=3.5 augmenter for the rest of the investigation was dropped. The L/D=2.5 A/B had both lower augmentation ratio and greater TSFC than the L/D=3.0 A/B.

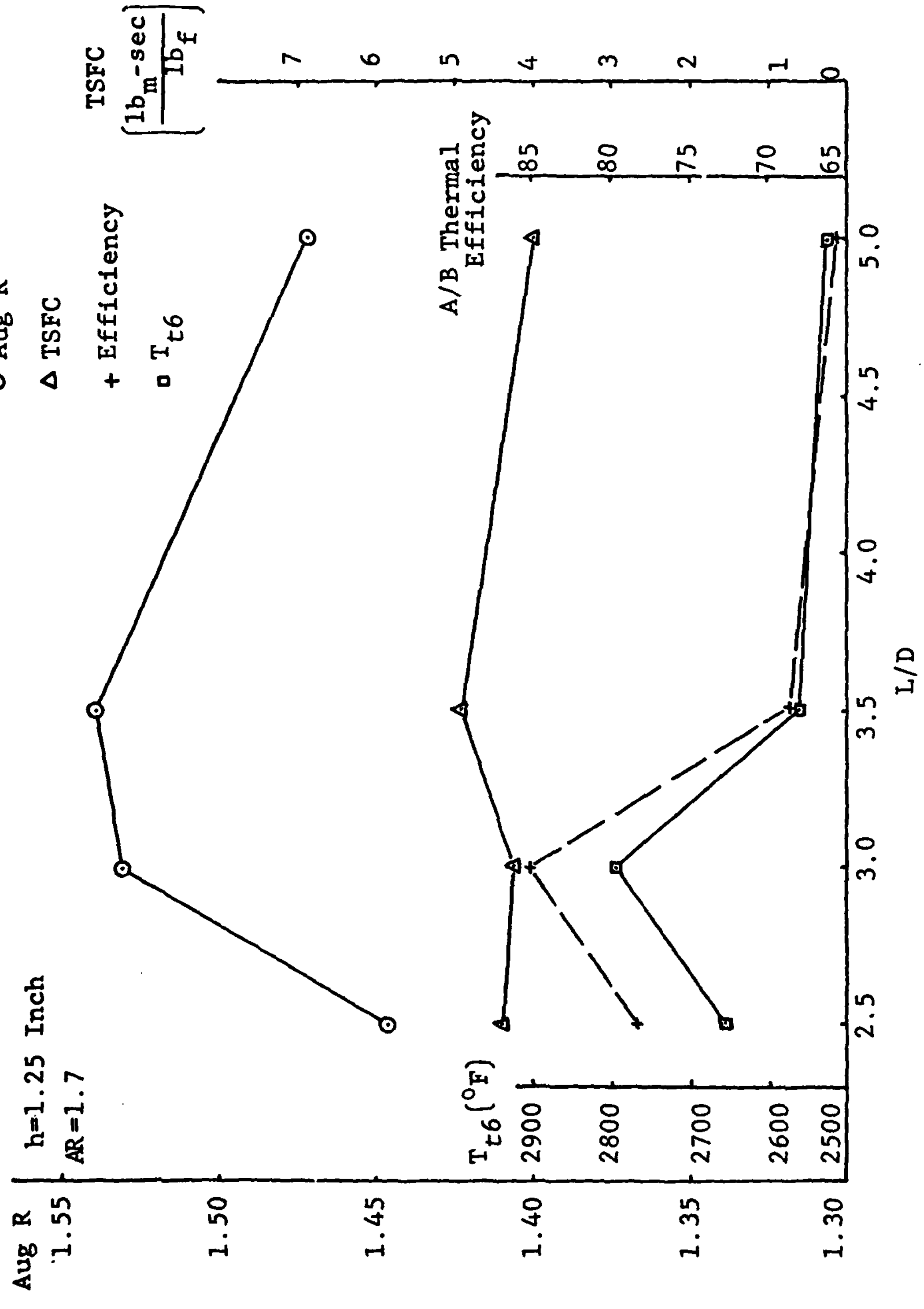


Fig. 9 Starting Point Performance for Various L/D

Figure 10 shows that TSFC and specific thrust are approximately the same for L/D=3 and L/D=5. Therefore the reduced weight of L/D=3 becomes a dominant factor.

Figure 11 shows starting point performance and best performance data points versus L/D, and it also shows the accompanying pressure losses. It can be observed that for L/D=3 augmentation ratio increases from start to max performance point, while pressure loss ($\Delta P_t/P_{t5}$) decreases. The same behavior is typical for L/D of 3.5, and the opposite behavior is observed in the case of L/D of 5.

In general, as can be seen from Fig. 8 the higher the average pressure loss for a certain configuration, the higher the thrust. But examining one cycle of A/B light-off and increase in fuel flow, it is seen that sometimes pressure loss increases at a certain fuel-air ratio, but thrust does not increase. This means that pressure loss is not the only variable that governs the thrust augmentation; burner efficiency as well as residence time of the fuel must also be considered. No one of the above mentioned variables by itself governs the thrust augmentation. It must be remembered that this investigation deals with an engine-A/B combination. It means that the gas supply source to the A/B is not rigid but is affected by change of conditions in the A/B. Increasing pressure loss may result in more active recirculation flow, but with an accompanying reduction in mass flow from the engine. As a result T_{t4}

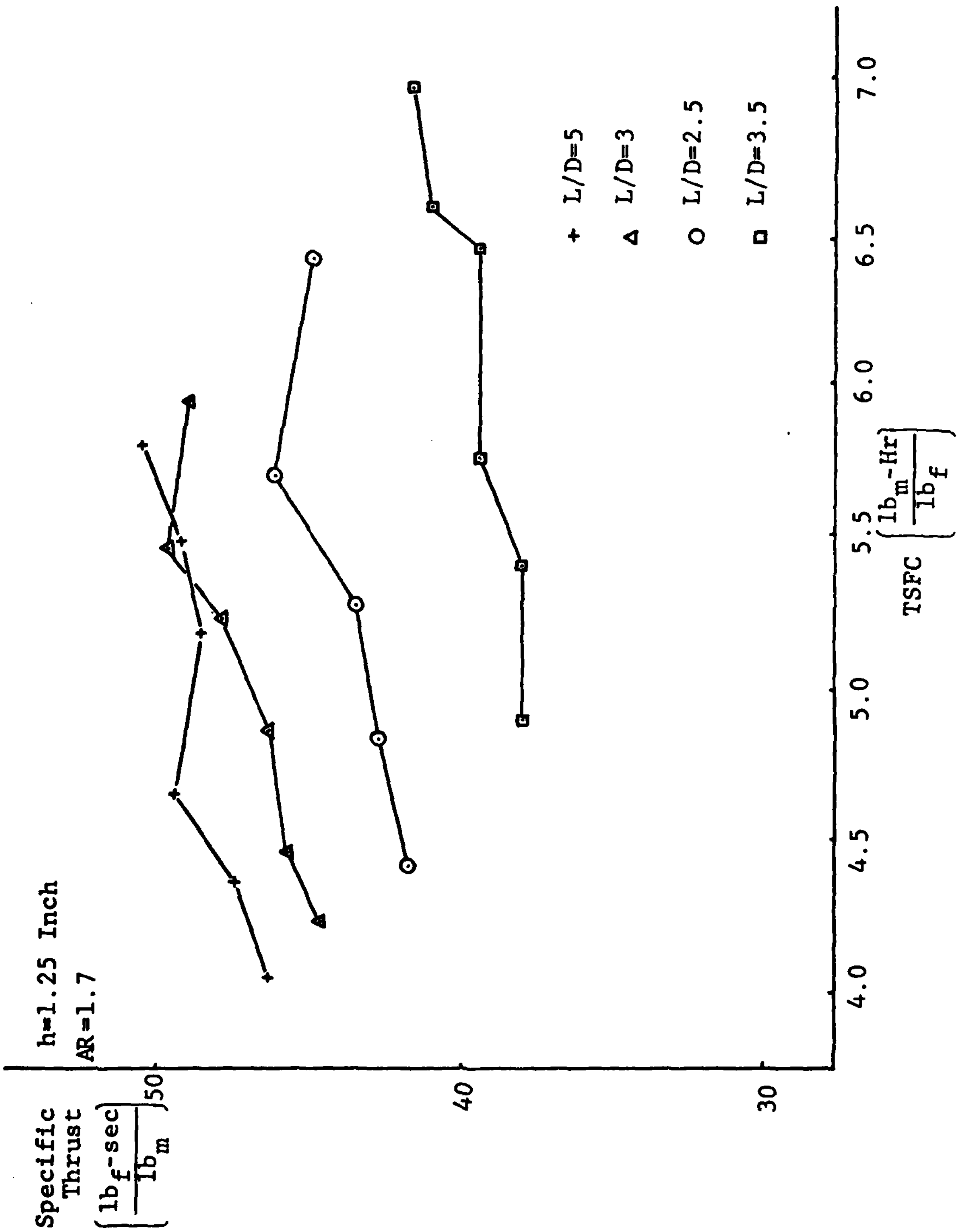


Fig. 10 Effects of L/D Variation on Specific Thrust and TSFC

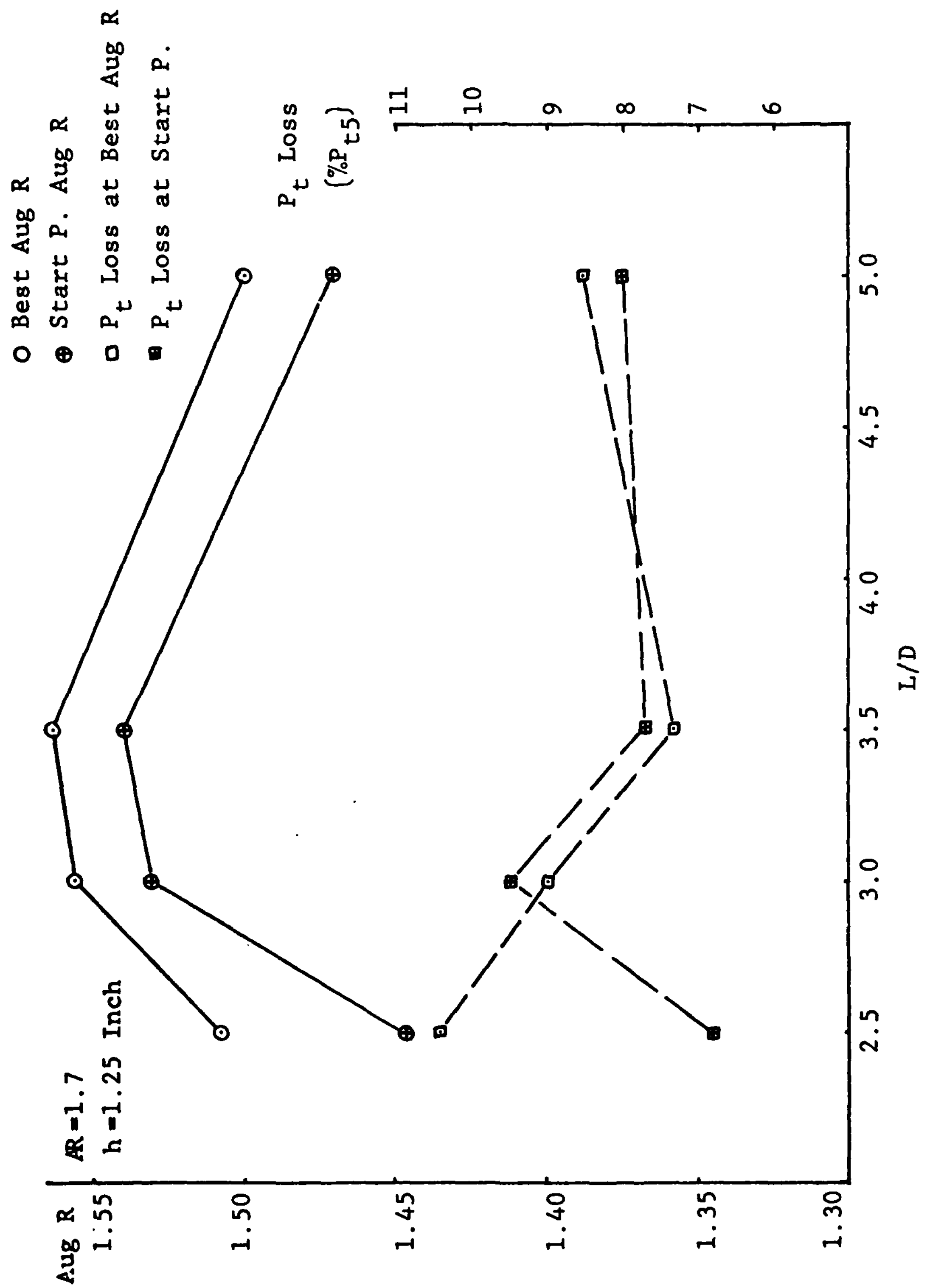


Fig. 11 Starting Point Performance vs Best Performance Point at Various L/D

will increase which requires engine retrimming in order to maintain constant T_{t4}, resulting in loss of overall thrust. Accordingly, high augmentation ratio may be accompanied by lower overall thrust as shown in Fig. 12.

In order to investigate the separate contribution of each component of the propulsion system (i.e. engine and augmenter) the connection between the augmenter and the engine must be such that augmenter thrust alone could be measured, and total thrust could be measured. This will enable an analysis of the thrust, in which component of the propulsion system it is produced and it's cost in terms of fuel consumption.

It was also noticed that when the engine had high MIL 1 thrust, the augmentation ratio was low as shown by Fig. 8. It may be that this phenomenon is typical for a particular engine. More definitive measurements especially at the turbine exit duct area are required before a suitable explanation can be given. These measurements should include fuel spray pattern, gas velocity, vaporization, fuel residence time before entering the A/B, and the amount of unburned air in the gas flow entering the A/B. In order to investigate the possibility of engine - A/B operation with a fixed A/B nozzle area (i.e. no ejectable nozzle) the A/B operation was intentionally stopped and restarted. This presented no difficulty in engine or A/B operation.

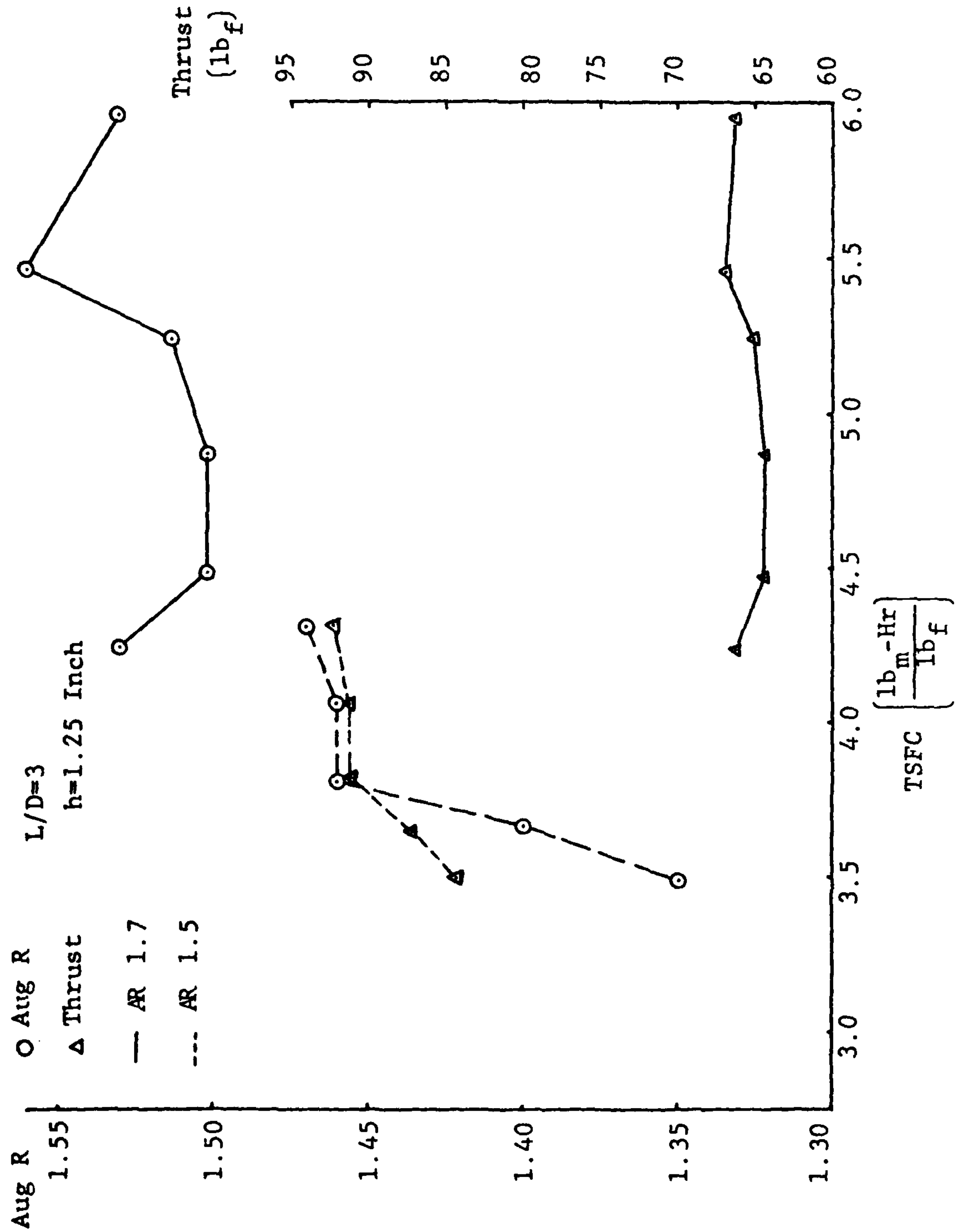


Fig. 12 Aug R and Thrust Relations at Various TSFC

Effect of Air Injection at the Step

The air injection methods and air injection directions which were tried did not increase thrust or augmentation ratio. Figures 13 and 14 show the best performance achieved by a particular type of air injection, and the performance of the propulsion system without the air injection. In the case with air injection augmentation ratio and thrust were reduced, TSFC increased, pressure loss decreased, but afterburner thermal efficiency increased. This shows that the air injection did help in providing better mixing and combustion. This also is reflected in higher T_{t6} temperature achieved (T_{t6}=2640F with air injection versus T_{t6}= 2580F without air injection, at the starting point of A/B operation). But the bleed air from the engine compressor, to be used for injection created a higher T_{t4} and the engine had to be retrimmed, which means that fuel flow to the engine had to be reduced in order to maintain T_{t4}=1700F. As a result engine speed was reduced air mass flow and overall propulsion system thrust decreased.

The air injection configuration that achieved the best performance was clockwise tangential injection at the inner diameter of the step, as shown in Fig. 4b, e, g. In this case the air injection was opposite to the turbine exit swirl.

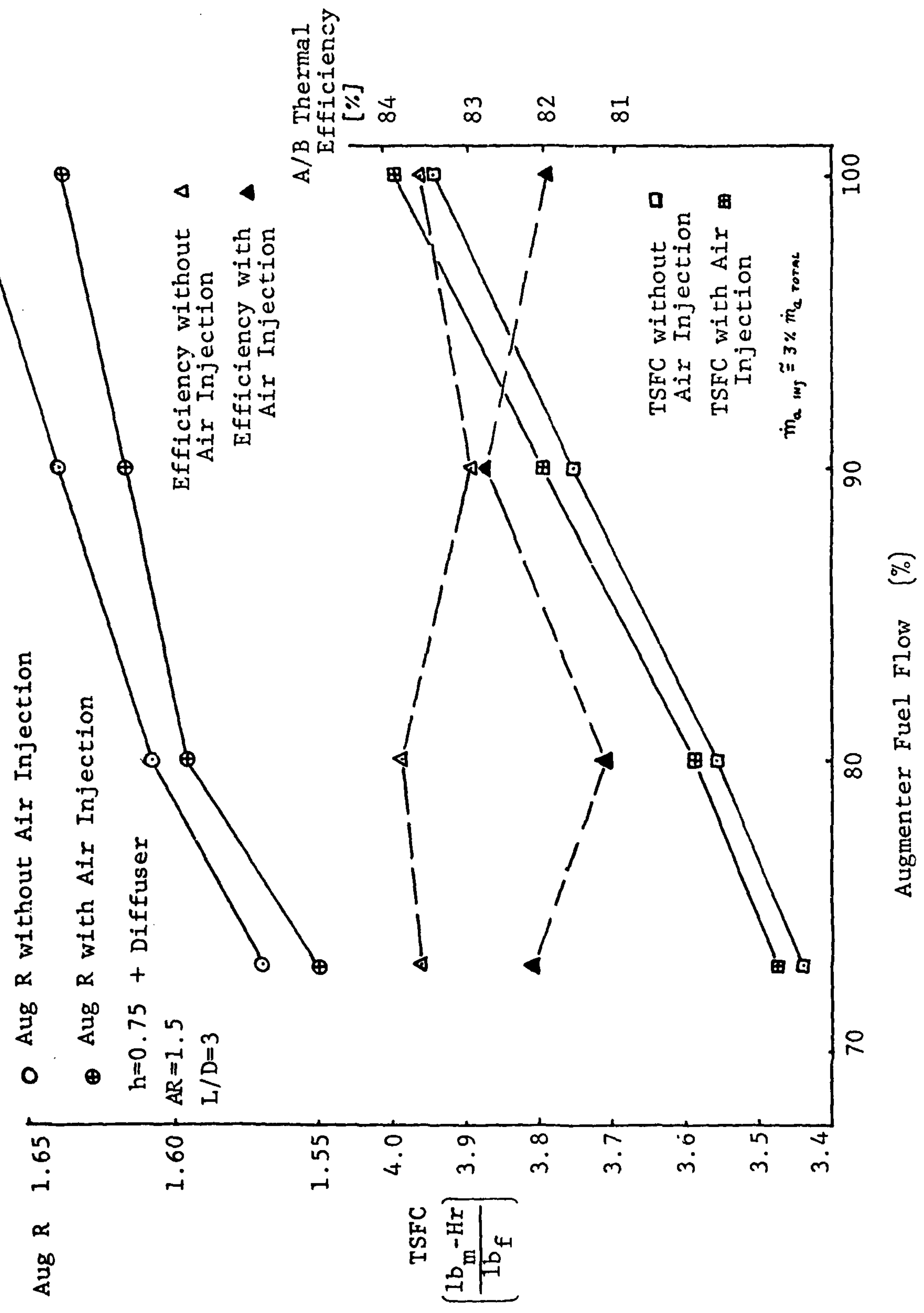


Fig. 13 Effects of Circumferential CW Air Injection on Aug R, Efficiency and TSFC

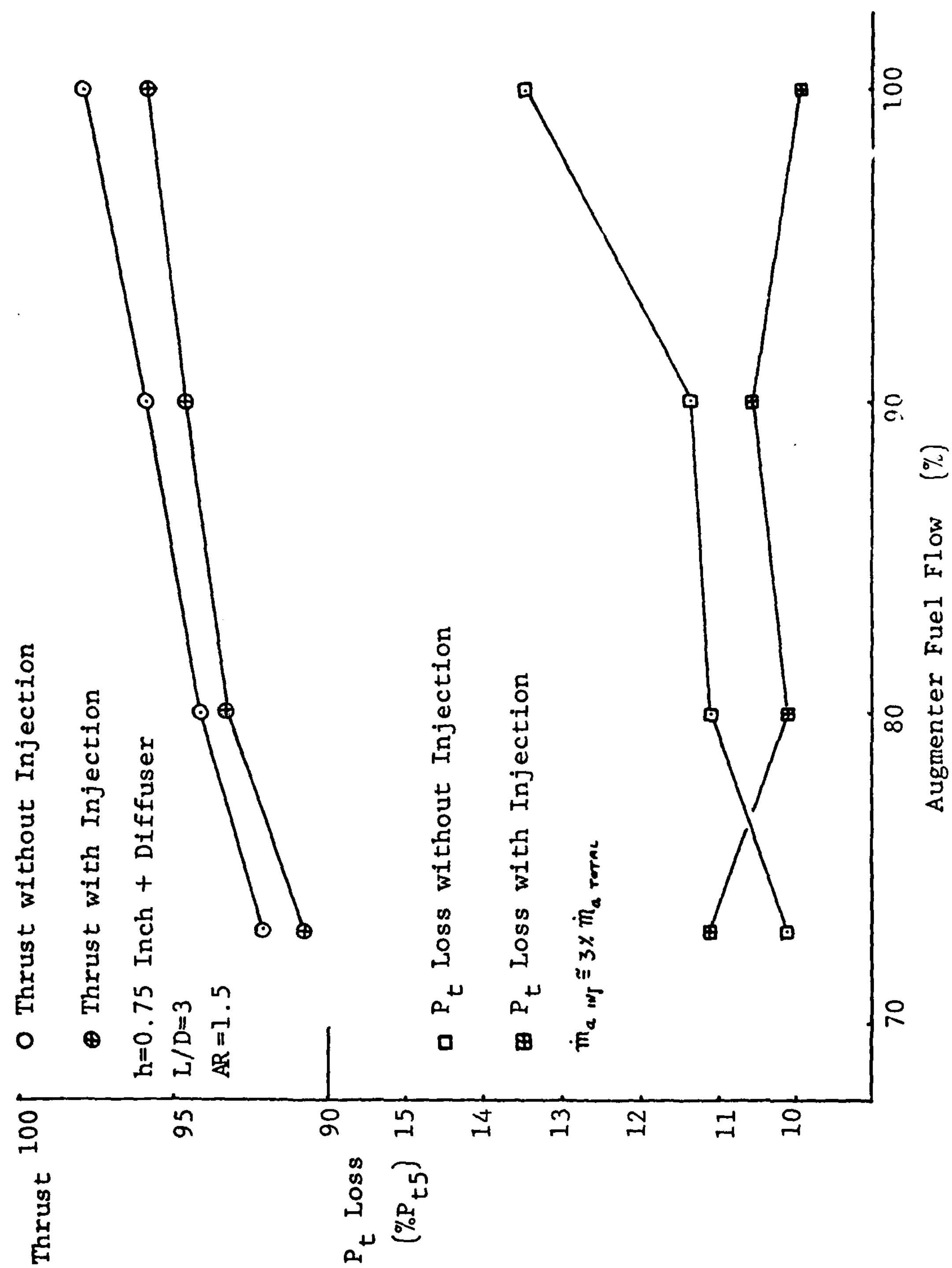


Fig. 14 Effects of Circumferential CW Air on Thrust and Pressure Loss

Eccentrically Mounted A/B

The effects of eccentrically mounted A/B on A/B light-off and stability of operation were studied. No difficulty was encountered with A/B light-off but higher initial fuel-air ratio was required. At L/D=3 with concentric mounting, the starting fuel-air ratio was 0.04781; in the off-center configuration starting fuel-air ratio was 0.0489. Other than that, the light-off and stability of operation were normal as in the concentric version.

At a fuel-air ratio of 0.0584 the noise level of the propulsion system increased significantly and stayed high up to the highest A/B operation point, where fuel-air ratio was 0.068. When decreasing the fuel flow to the A/B the high noise stopped immediately after passing the same fuel-air ratio (0.0584).

Figures 15 and 16 show that at TSFC higher than 4.0 the augmentation ratio of the off-center configuration is higher, and the specific thrust is nearly the same when compared with the concentric configuration. Pressure loss is 1% higher on the average in the off-center configuration.

Some concern has been expressed (Ref.12) about encountering combustion instability with off-center flow into the A/B, perhaps resulting in flame wash out. No indication of this was observed.

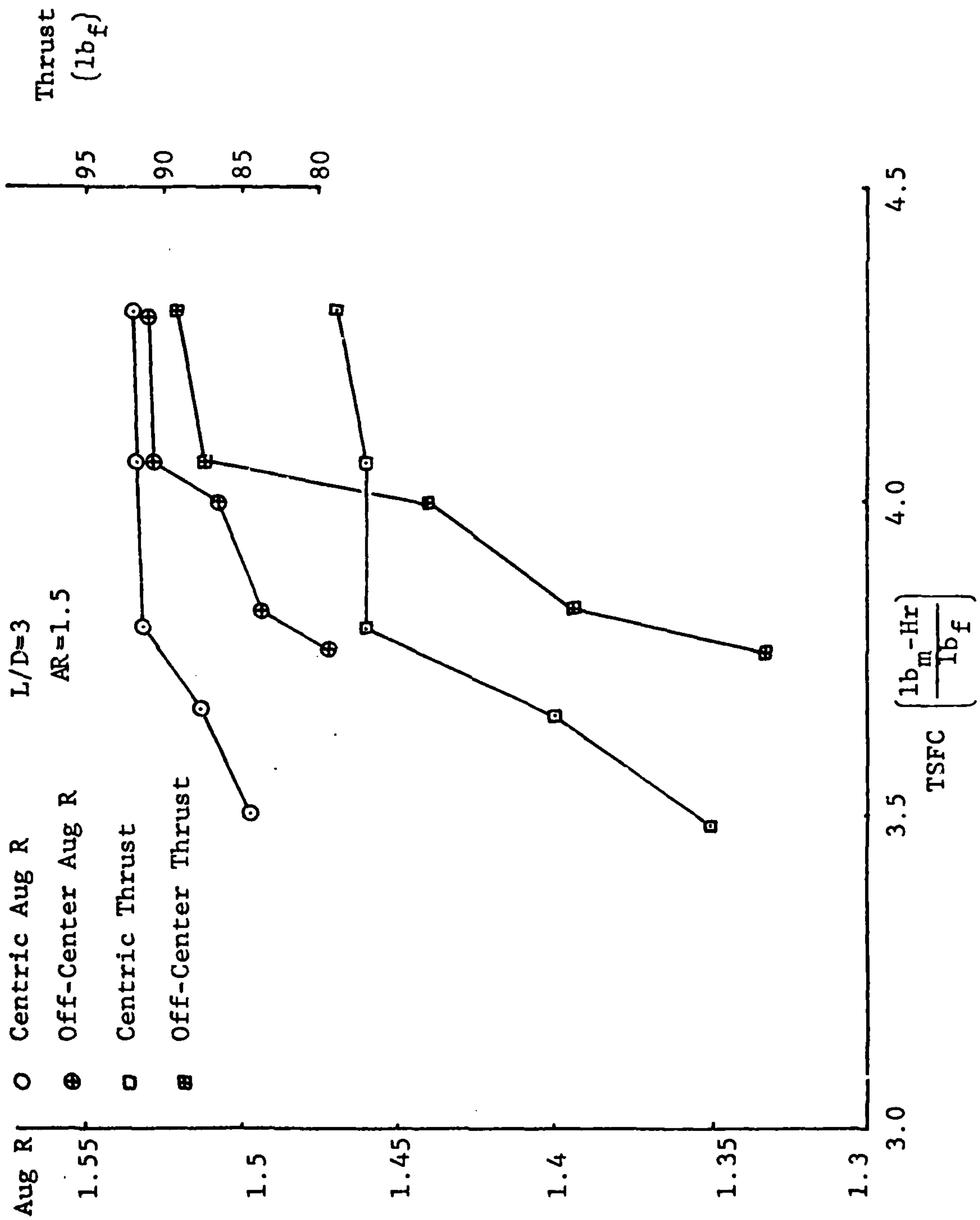


Fig. 15 Effects of Eccentrically Mounted
A/B on Aug R and Thrust

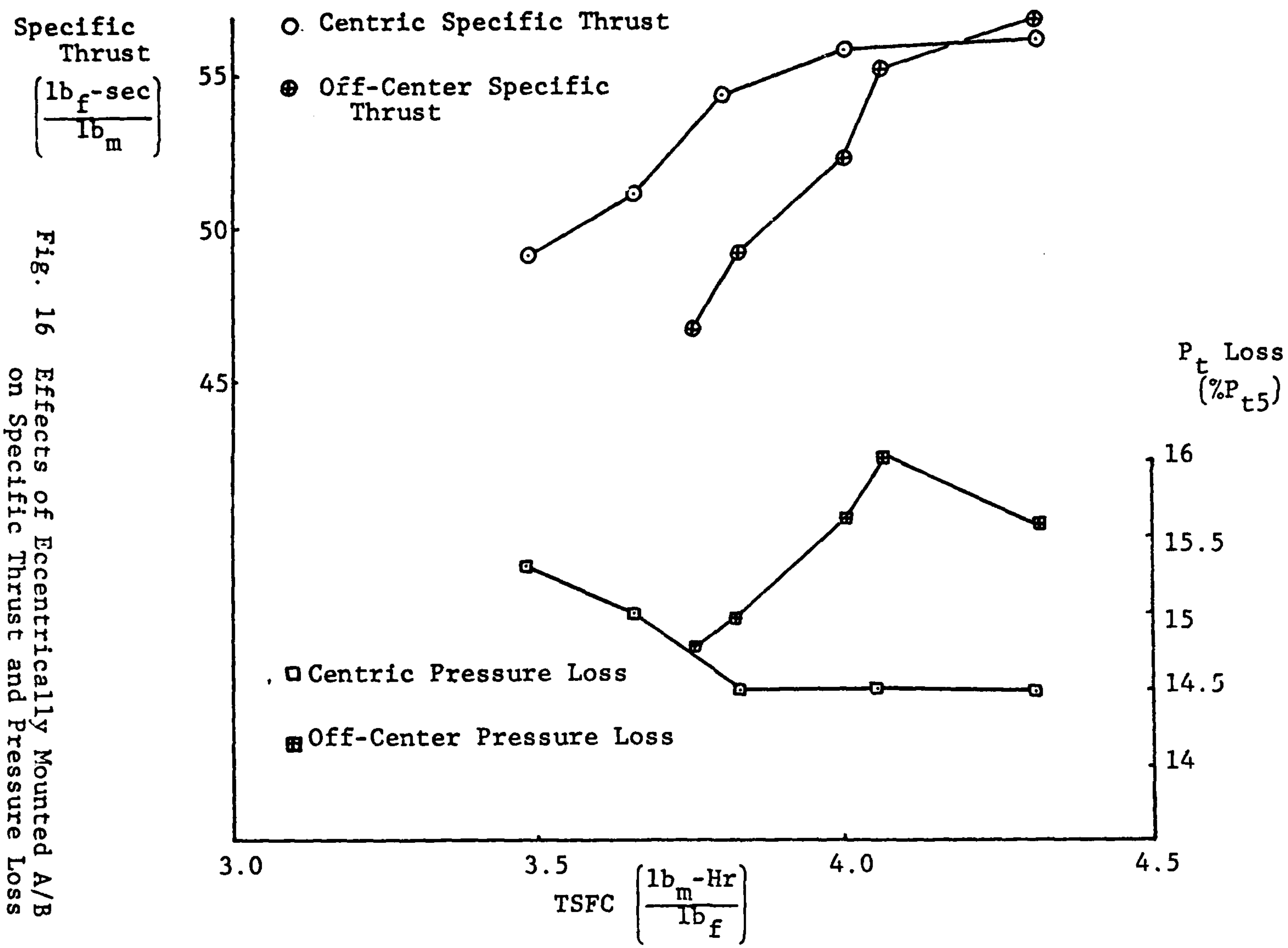


Fig. 16 Effects of Eccentrically Mounted A/B on Specific Thrust and Pressure Loss

V. Conclusions and Recommendations

Conclusions

Dump combustor thrust augmenters of L/D=2.5 to L/D=5, and augmenter configurations having air injection and off-center mounting, were operated successfully. The performance variation in this L/D range and the various A/B configurations has led to the following conclusions and recommendations:

1. An afterburner having L/D=3 has performance similar to that of an afterburner having L/D=5 and is 40% lighter. Augmenters having L/D=2.5 and L/D=3.5 have decreased performance in comparison with augmenter of L/D=3.

2. The augmenter should be operated on a basis at on-off operation mode, at the lowest TSFC or fuel-air ratio possible. This will save fuel flow regulating device, weight and fuel.

3. A/B shut-off and relight is possible without need of an adjustable area nozzle.

4. Air injection into the step zone produced decreased performance. In addition, weight, cost and complication make such a system undesirable.

5. A/B off-center mounting operation is possible and does not present any difficulties. The decrease in

performance in this case is negligible especially at TSFC higher than 4.0.

6. Whenever such a system of engine-A/B combination is analyzed, augmentation ratio should be considered together with MIL 1 thrust and total thrust. Augmentation ratio alone does not represent the whole performance.

Recommendations

1. The future investigations should be performed with L/D=3.

2. The optimal A/B nozzle design for A/B on-off operation should be studied and designed.

3. The possibility of using compressor bleed air for turbine housing and A/B cooling in a light weight design should be studied.

4. Engine combustion chamber connections to compressor and turbine housings should be redesigned in order to minimize frontal area and increase thrust to weight ratio.

5. An integral automatic starting system, that includes an acceleration control device and T_{t4} protecting mechanism should be designed and tested.

6. Integral fuel and lubrication systems suitable for flight operation should be developed.

Bibliography

1. Barham, Robert W. An Investigation of the Effects of Flameholding and L/D on the Performance of a Small Turbojet with a Dump Combustor Afterburner. Thesis. Wright-Patterson AFB, Ohio: Air Force Institute of Technology, March 1978.

2. Chowdhury, P. Roy. Interaction of Turbulent Air Jets with an Impinging Reacting Stream-An Application to an Advanced Air Breathing Propulsion System. Paper. University of Southern California, August 1977.

3. CPIA Publication 276. Recommended Ramburner Test Reporting Standards. John Hopkins University, March 1976.

4. Drewry, James E. Fluid Dynamic Characterization of Sudden Expansion Ramjet Combustor Flowfields. Presented as Paper 77-203 at AIAA 15th Aerospace Science Meeting: Los Angeles, California, January 24-26, 1977.

5. Durniak, J. D. Radical Designs for Low Cost Expendable Turbojets. Thesis. Wright-Patterson AFB, Ohio: Air Force Institute of Technology, March 1977.

6. Elrod, William C. Review of Combustion Efficiency Determination Methods in Air Augmented Rocket Propulsion. Paper RPMR-70-007. Wright-Patterson AFB, Ohio:Air Force Institute of Technology, August 1970.

7. Hojnacki, John T. Ramjet Engine Fuel Injection Studied. Air Force Aero-Propulsion Lab Report No. AFAPL-TR-72-76. Wright-Patterson AFB, Ohio, November 1972.

8. Kent, T. J. Conceptual Study of a Low Cost Turbojet Engine. Thesis. Wright-Patterson AFB, Ohio: Air Force Institute of Technology, March 1976.

9. MacInnes, Hugh. Turbochargers. Library of Congress Catalog No. 76-6002. June 1976.

10. Ortwerth, P. J.; J. T. Hojnacki. Liquid Ramjet Dump Combustor Development. Paper. Wright-Patterson AFB, Ohio: Air Force Propulsion Lab.

11. Stull, F. D.; R. R. Craig; J. T. Hojnacki. Dump Combustor Parametric Investigation. Paper. Wright-Patterson AFB, Ohio: Air Force Propulsion Lab.

12. Wilkinson, David B. Personal Discussions. Wright-Patterson AFB, Ohio: Air Force Propulsion Lab, June 1978.

13. Wolfe, Byron L. Thrust Augmentation for a Low Cost Turbojet by Means of Sudden Expansion Afterburner. Thesis. Wright-Patterson AFB, Ohio: Air Force Institute of Technology, March 1977.

Appendix A

Data Reductions

Data for this investigation was reduced in the same general way as that used by Wolfe and Barham (Refs. 13, 1). The equations used are reproduced here for convenience to the reader.

Recorders

A Honeywell 906B visicorder was used to record various pressures, temperatures, and thrust. Table II shows the appropriate scale factors used on the visicorder and on which side the zero setting was located. Prior to calibration, all channels except 6 and 10 (T_{t6} and T_3) were zeroed on the appropriate side. Channels 6 and 10 were set to control room temperature.

Table II
Scale Factors

Quantity	Channel	Zero Setting	Scale Factor
T_{t6}	5	C/R Temp-Left	800F/inch
P_3	7	Right	8psi/inch
P_4	8	Left	5psi/inch
P_5	9	Right	10psi/inch
T_3	10	C/R Temp-Left	100F/inch
F	11	Right	25lb/inch
P_{t5}	12	Right	5psi/inch

Main engine and afterburner fuel flows were recorded on a Honeywell Chart Recorder Model SY153X(28). This unit contained a frequency-to-voltage converter for each channel and displayed frequency as a linear output deflection proportional to the input frequency. The actual fuel flow was then computed using the formula given under the section titled Fuel Flow Rate.

Air Flow Rate

Air flow rate was determined by manually recording the ΔP measurement in inches of water directly from the water manometer. This measurement was then converted to mass flow by using the Bellmouth curve developed by Kent (Ref. 8). The mass flow was corrected to sea level standard conditions by using the equation:

$$m_a = \tilde{m}_a \frac{\sqrt{\theta}}{\delta}$$

where

$$\theta = \frac{T_o}{519^{\circ}R}$$

and

$$\delta = \frac{P_o}{29.92 \text{ in Hg}}$$

Fuel Flow Rate

Fuel flow was calculated by measuring the pen deflection

of the Honeywell recorder in percent and then inserting that value into the following equations:

$$\bar{\dot{m}}_f = (2.5 \times 10^{-4})(FF_e\% \times 3)\ \frac{SpGr}{0.75}$$

$$\bar{\dot{m}}_{a/b} = (2.5 \times 10^{-4})(FF_{a/b}\% \times 3)\ \frac{SpGr}{0.75}$$

where SpGr=fuel specific gravity.

The total uncorrected fuel flow was calculated using the equation:

$$\bar{\dot{m}}_{ft} = \bar{\dot{m}}_{fe} + \bar{\dot{m}}_{a/b}$$

Sea level standard condition corrections were done by applying the equation:

$$\dot{m}_{ft} = \frac{\bar{\dot{m}}_{ft}}{\delta\sqrt{\theta}}$$

Fuel Air Ratio

The corrected values for fuel/air ratio were obtained by applying the simple equation:

$$f_t = \dot{m}_{ft}/\dot{m}_a$$

Compressor Pressure Ratio

The compressor pressure ratio was computed using the equation:

$$CPR = \frac{P_3+P}{P}$$

where P_3 = compressor pressure in psig.

Thrust

Thrust was corrected to standard sea level conditions by using the equation:

$$F = \bar{F}/\delta$$

Thrust Specific Fuel Consumption

Corrected values of TSFC were obtained using the formula:

$$TSFC = (\dot{m}_{ft} \times 3600)/F$$

Specific Thrust

Specific thrust at standard sea level conditions was computed using:

$$F_{sp} = F/\dot{m}_a$$

RPM

RPM readings were obtained by manually recording the digital counter reading and multiplying it by a factor of 60 seconds/minute. It was then corrected to standard conditions by using the formula:

$$N = \bar{N}/\sqrt{\theta}$$

A/B Thermal Efficiency

Thermal efficiency was estimated using the relationship:

$$\eta_T = \frac{(T_{t5}-T_{t6})_{act}}{(T_{t5}-T_{t6})_{ideal}}$$

which is found in Ref. 7. Because of low mach numbers in the engine-A/B (Ref. 5), T_5 and T_6 were assumed close to T_{t5} and T_{t6} for the efficiency calculation. According to Ref. 7, T_{t5} actual is equal to T_{t5} ideal, therefore, the gage reading of T_5 was used for both. The T_6 visicorder reading was converted using the scale factor and used for T_{t6} actual. The value for T_{t6} ideal was chosen from Ref. 5, which gives adiabatic flame temperature based on fuel-air ratio and compressor discharge temperature, T_3.

Augmentation Ratio

It must be pointed out that augmentation ratio was calculated by dividing afterburner thrust by the thrust at

MIL 1. It can also be calculated using MIL 2 values. This distinction must be kept in mind when comparing these results with other studies.

Air Injection Flow Rate

Assuming the injected air Mach number is less than 0.3 (since the highest Mach number in the engine flow path is 2.7 in the A/B exit nozzle) the injected air mass flow calculation is based on the non-compressible flow rate formula for a step orifice:

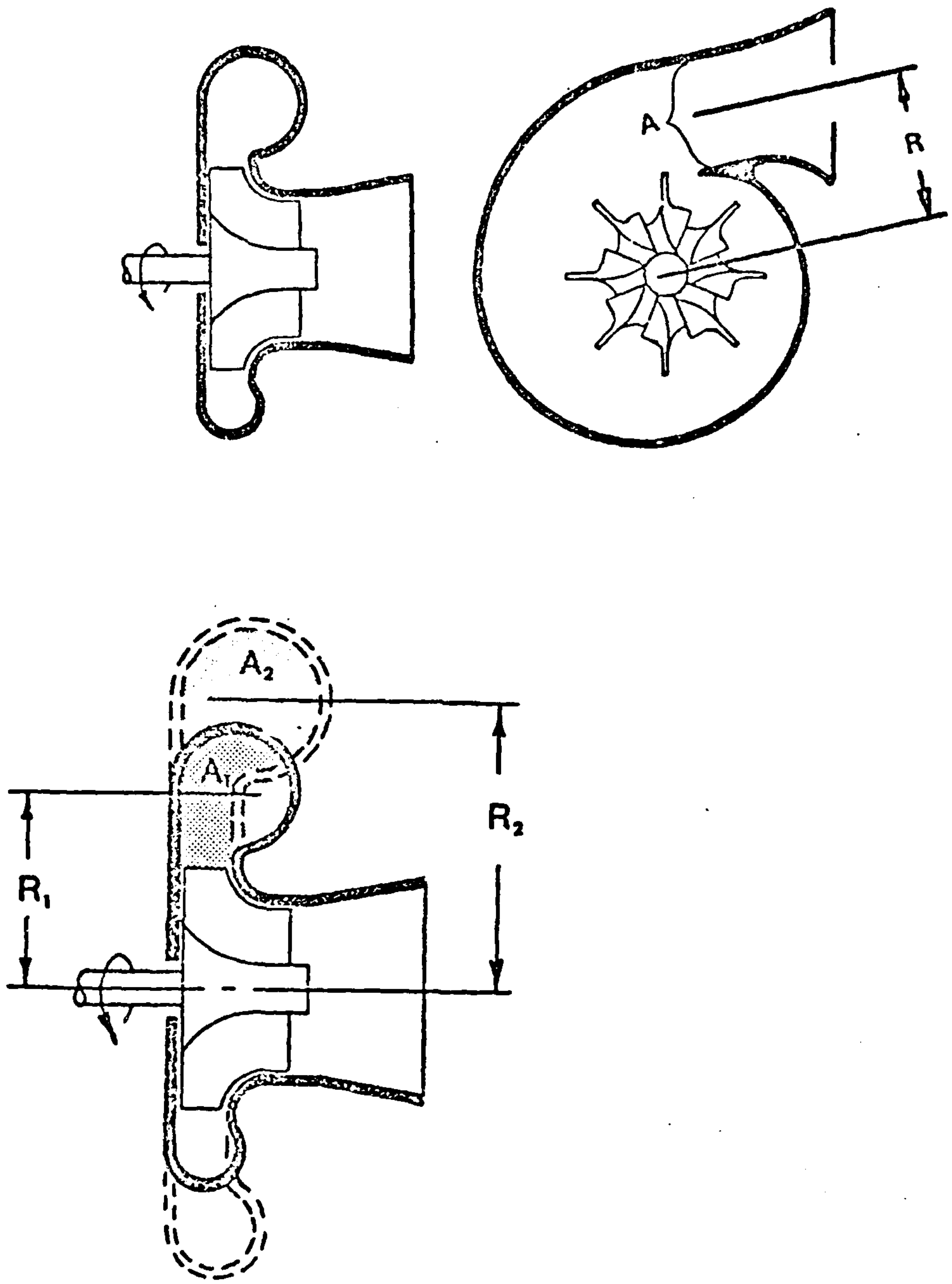
$$Q = N\, C_d\, A \sqrt{\frac{2\Delta P}{\zeta}}$$

where
- Q - flow rate
- C_d - discharge coefficient (in this case N0.62)
- A - orifice area
- ΔP - pressure drop through orifice
- ζ - density
- N - number of injection orifices.

A/B Total Pressure Loss

$$\frac{P_{t5}-P_{t6}}{P_{t5}} \times 100$$

Aspect Ratio



Area A. Divided by R (the Distance Measured from the Center of the Turbine Wheel to the Centroid of Area A), is a Measure of Sizing Turbine Housings. A Larger A/R Ratio Slows the Turbine. A Small A/R Ratio Increases the Speed of the Turbine. Note A/R_1 and A/R_2 in the Above Diagram have the same A/R Ratio.

Fig. 17 Turbine Housing Aspect Ratio

Vita

Rami Dotan was born on August 1, 1945 in Bucharest, Rumania. In 1950 he immigrated to Israel. He completed high school studies in Holon in 1963 and was drafted into the Army on August 1, 1963. After he was admitted to college, the Technion (Israeli Institute of Technology) in Haifa, on October 1963, he studied in the faculty of mechanical engineering, and in the same time served at Reserve Officer Training Corps. In 1966 he completed his officer training in the Army Officers School. He participated in 1967 war as an infantry officer. In 1968 he received a degree in Mechanical Engineering and went back to the Army. In 1969 he was transferred to the Air Force and served in technical and command jobs. He participated in 1973 war as an Air Force Maintenance Officer, Commanding Officer of the Jet Engine Depot Shop. He entered the Air Force Institute of Technology in June 1977. He is married to Michal, and has two boys.

Permanent Address: 19 Krauze St.
Holon, Israel

Unclassified

SECURITY CLASSIFICATION OF THIS PAGE (When Data Entered)

REPORT DOCUMENTATION PAGE

READ INSTRUCTIONS BEFORE COMPLETING FORM

1. REPORT NUMBER: AFIT/GAE/AA/78D-4
2. GOVT ACCESSION NO.:
3. RECIPIENT'S CATALOG NUMBER:
4. TITLE (and Subtitle): TURBOJET THRUST AUGMENTATION BY SUDDEN EXPANSION TYPE AFTERBURNER
5. TYPE OF REPORT & PERIOD COVERED: M.S. Thesis
6. PERFORMING ORG. REPORT NUMBER:
7. AUTHOR(s): Rami Dotan, Major IAF
8. CONTRACT OR GRANT NUMBER(s):
9. PERFORMING ORGANIZATION NAME AND ADDRESS: Air Force Institute of Technology(AFIT-EN), Wright-Patterson AFB,Ohio 45433
10. PROGRAM ELEMENT, PROJECT, TASK AREA & WORK UNIT NUMBERS:
11. CONTROLLING OFFICE NAME AND ADDRESS:
12. REPORT DATE: December 1978
13. NUMBER OF PAGES: 55
14. MONITORING AGENCY NAME & ADDRESS (if different from Controlling Office):
15. SECURITY CLASS. (of this report): Unclassified
15a. DECLASSIFICATION DOWNGRADING SCHEDULE:
16. DISTRIBUTION STATEMENT (of this Report): Approved for public release; distribution unlimited.
17. DISTRIBUTION STATEMENT (of the abstract entered in Block 20, if different from Report):
18. SUPPLEMENTARY NOTES:

Approved for public release;IAW AFR 190-17
Joseph P. Hipps, Major, USAF
Director of Information

19. KEY WORDS (Continue on reverse side if necessary and identify by block number):
20. ABSTRACT (Continue on reverse side if necessary and identify by block number)

An axisymmetric sudden expansion afterburner coupled to a small turbojet engine in various configurations, has been studied. The effects of length to diameter ratio variation on performance were investigated. The maximum thrust achieved was 62 pounds without afterburner augmentation, and 92 pounds with afterburner augmentation, each with turbine inlet temperature of 1700F.

Various methods of air injection into the combustion zone of the afterburner were examined, and found to be ineffective (cont'd)

DD FORM 1 JAN 73 1473 EDITION OF 1 NOV 65 IS OBSOLETE

Unclassified

SECURITY CLASSIFICATION OF THIS PAGE (When Data Entered)

(20. Cont'd) for purpose of thrust augmentation.

An eccentrically mounted afterburner with an eccentricity of 1.25 inch has been operated successfully. The performance of the propulsion system decreased in this configuration only by a negligible amount.